NACA

# RESEARCH MEMORANDUM

## A PRELIMINARY INVESTIGATION OF THE USEFULNESS OF CAMBER IN OBTAINING FAVORABLE AIRFOIL-SECTION DRAG CHARACTERISTICS AT SUPERCRITICAL SPEEDS

By Gerald E. Nitzberg, Stewart M. Crandall, and Perry P. Polentz

Ames Aeronautical Laboratory
Moffett Field, Calif.



NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON
October 7, 1949

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia
23665

NASA

JUN 1 6 1983

Reply to Attn of 139A

TO: Distribution

FROM: 180A/Security Classification Officer

SUBJECT: Authority to Declassify NACA/NASA Documents Dated Prior to January 1, 1960

Effective this date, all material (informal, correspondence) classified by this Center prior to January 1, 1960, is declassified. This action does not include material derivatively classified at the Center upon instructions from other Agencies.

Immediate re-marking is not required; however, until material is re-marked by lining through the classification and annotating with the following statement, it must continue to be protected as if classified:

> "Declassified by authority of LaRC Security Classification Officer (SCO) letter dated June 16, 1983," and the signature of person performing the re-marking.

If re-marking a large amount of material is desirable, but unduly burdensome, custodians may follow the instructions contained in NHB 1640.4, subpart F, section 1203.604, paragraph (h).

This declassification action complements earlier actions by the National Archives and Records Service (NARS) and by the NASA Security Classification Officer (SCO). In Declassification Review Program 807008, NARS declassified the Center's "Research Authorization" files, which contain reports, Research Authorizations, correspondence, photographs, and other documentation. Earlier, in a 1971 letter, the NASA SCO declassified all NACA/NASA formal series documents with the exception of the following reports, which must remain classified:

| Document No. | First Author |
|---|---|
| E-51A30 | Nagey |
| E-53G20 | Francisco |
| E-53G21 | Johnson |
| E-53K18 | Spooner |
| SL-54J21a | Westphal |
| E-55C16 | Fox |
| E-56H23a | Himmel |

JUN 2 3 1983

If you have any questions concerning this matter, please call Mr. William L. Simkins at extension 3281.

Jess G. Ross
2898

Distribution:
SDL 031

cc:
NASA Scientific and Technical
Information Facility
P.O. Box 8757
BWI Airport, MD 21240

NASA--NIS-5/Security
180A/RIAD
139A/TU&AO
6-15-83
139A/WLSimkins:elf 06/15/83 (3281)
139A/JS 6-15-83

31-01 HEADS OF ORGANIZATIONS
HESS, JANE S.
MAIL STOP 185 BLDG 1194

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

# A PRELIMINARY INVESTIGATION OF THE USEFULNESS OF CAMBER IN OBTAINING FAVORABLE AIRFOIL-SECTION DRAG CHARACTERISTICS AT SUPERCRITICAL SPEEDS

By Gerald E. Nitzberg, Stewart M. Crandall,
and Perry P. Polentz

## SUMMARY

An investigation was made to determine the possibility of delaying at moderate or large lift coefficients the onset of the abrupt supercritical drag rise of an airfoil section by the use of camber. An analysis of the data from previous experimental studies, supplemented by calculations of drag-divergence characteristics of two basic thickness forms in combination with a variety of mean lines, indicates that significant gains in high-speed drag characteristics are to be obtained by cambering some airfoil sections. It was found experimentally, as predicted, that for an NACA 0010 airfoil section marked increases in drag-divergence Mach number are obtained by cambering the airfoil for a design lift coefficient of 0.3 with an NACA $a = 1.0$ mean line.

## INTRODUCTION

Early attempts to design airfoil sections which were advantageous for high-speed applications were based on considerations of the critical Mach number. The critical Mach number of a cambered airfoil section is usually higher, at moderate lift coefficients, that that of the symmetrical basic thickness form. However, when experimental data were obtained, it was found that the addition of camber led to adverse effects on the high-speed lift and pitching-moment characteristics and, in some cases, on the drag characteristics too. Because of these unpromising results no systematic study has been made of the effects of camber on high-speed force characteristics of airfoil sections.

Reference 1 shows that the drag-divergence Mach number, the Mach number at which the abrupt supercritical drag rise begins, provides a more useful criterion than critical Mach number for the study of the effects of shape changes on airfoil-section characteristics at high speeds. This reference also suggests a method for calculating drag-divergence Mach number.

The purpose of this investigation is to determine the possibility of increasing, at moderate or large lift coefficients, the drag-divergence Mach number of airfoil sections by the use of camber. Data of previous experimental investigations were examined and an analysis, using a method suggested by reference 1, was then made of the effect of a systematic variation of mean line on the drag-divergence Mach number of two basic thickness forms. As a result of this analysis a cambered airfoil section was designed which was expected to have higher drag-divergence Mach number at moderate lift coefficients than the basic thickness form. In order to determine whether the anticipated gains were realized and what effect this shape change had on other high-speed force characteristics, lift, drag, and pitching-moment data were obtained for the two (symmetrical and cambered) airfoil sections in the Ames 1- by 3-1/2-foot high-speed wind tunnel. These data are compared with experimental values for the NACA 64A010 airfoil section, which is generally considered to possess good high-speed characteristics.

## NOTATION

$a$ mean-line designation, fraction of chord from leading edge over which design load is uniform

$a_o$ lift-curve slope, per degree

$\alpha_o$ angle of attack, degrees

$c$ airfoil chord

$c_d$ section drag coefficient

$c_l$ section lift coefficient

$c_{l_i}$ design section lift coefficient

$c_{m_{c/4}}$ section pitching-moment coefficient about quarter-chord point

$M$ free-stream Mach number

$M_d$ drag-divergence Mach number (Mach number at which slope of curve of drag coefficient versus Mach number attains a value of 0.10.)

$P$ pressure coefficient $\left(\frac{p-p_o}{q_o}\right)$

$p$ local static pressure

$p_o$ free-stream static pressure

$P_R$ resultant pressure coefficient, difference between local upper- and lower-surface pressure coefficients

$q_o$ free-stream dynamic pressure

$x$ distance along chord

$y$ distance perpendicular to chord

$y_c$ mean-line ordinate

## ANALYSIS

### Analysis of Experimental Data

Plots of the drag-divergence Mach number versus lift coefficient are presented in figure 1 for a number of cambered airfoil sections derived from four NACA basic thickness forms. The experimental data presented in figure 1 were obtained from references 2, 3, and 4 and include substantially all the available experimental information applicable to the problem. The calculated curves shown in this figure will be discussed later. For the convenience of the reader the types of thickness distributions and mean-line shapes considered throughout the report are presented in figure 2.

The values of drag-divergence Mach number presented in figure 1 were determined as the free-stream Mach number at which the slope of the curve of drag coefficient against Mach number for a constant angle of attack has a value of 0.1. Ordinarily this definition provided a good measure of the free-stream Mach number at which the drag coefficient ceased to be essentially independent of Mach number and began to increase abruptly. However, for data obtained at Reynolds numbers below about 2 million it was often found that the increase in drag coefficient took place gradually instead of abruptly. For such data, the concept of drag divergence occurring at a definite Mach number has less utility, and conclusions based thereon must be made with less assurance. Flagged symbols are used in the figures to indicate results of this nature.

The data of figure 1 show that, except for the thin NACA 16-006 series airfoil sections, the addition of camber had little or no beneficial effect on the drag-divergence Mach number at moderate lift coefficients (0.2 to 0.4). In the majority of cases, however, it is to be noted that significant gains were obtained at high lift coefficients. If this gain could be extended to lower values of lift coefficient, it would be advantageous to do so.

### Analysis Utilizing Calculated Data

To investigate the possibility of extending the useful range of camber to lower lift coefficients, the method suggested in reference 1 was used to compute the effect on drag-divergence Mach number of a systematic variation of camber. A brief review of the theory of this reference and a numerical example illustrating the method are presented in an appendix to this report. In studying the effect of camber on the drag-divergence Mach number, it is important to make comparisons on the basis of equal lift coefficients rather than equal angles of attack. This required introducing a slight variation of the method of reference 1. Calculated values of drag-divergence Mach number compared with experimental values in figure 1 indicate that this calculation procedure leads to results which are in substantial agreement with experiment.

The effects of several types of camber lines, in combination with the NACA 0010 and 64A010 thickness distributions, on the calculated variation of drag-divergence Mach number with lift coefficient are considered in three categories: (1) Effects produced by different types of chordwise distribution of camber, each with approximately the same position of maximum camber; (2) effects of marked variation of the position of maximum camber for camber lines with equal design lift coefficients; and (3) effects of different amounts of camber for a given type of mean line. Each of these will be discussed in turn.

The calculated drag-divergence Mach numbers for the two airfoil sections to which have been applied the NACA $a = 1.0$, 65, $a = 0.4$, and 64-type camber lines are presented in figure 3. As may be seen in figure 2, the first two of these camber lines have the position of maximum camber at 50-percent-chord station, and the latter two at 40 percent. In each case the amount of camber corresponds to a design lift coefficient of 0.3. The principal observation to be made from figure 3 is that, for a given basic thickness form and a given chordwise location of the point of maximum camber, the choice of the particular camber line to be used appears to be of some importance but is secondary to effects of other camber variations discussed later. It is also interesting to note that, although application of camber to the NACA 64A010 basic thickness form provides no gain at lift coefficients smaller than about 0.4, application of camber to the NACA 0010 basic thickness form provides distinct improvement of the drag-divergence Mach number for lift coefficients larger than about 0.1.

The effect of varying the position of maximum camber for camber lines with design lift coefficients of 0.3 is indicated in figure 4. The mean lines used in this comparison are the NACA 62, 65, and 68 types. It is noted (fig. 4(a)) that the NACA 68-type camber line (maximum camber at 80-percent chord) offers more improvement in drag-divergence Mach number than does either the NACA 65 or 62 camber line (maximum camber at 50-percent and 20-percent chord, respectively). Figure 4(b) indicates that with rearward shift of the position of maximum camber there is an increase in drag-divergence Mach number at lift coefficients greater than about 0.2 and a decrease in drag-divergence Mach number at smaller coefficients for the NACA 0010 basic thickness form.

The effect of varying the amount of camber, for the $a = 1.0$ type of camber line, is shown in figure 5 for these same two basic thickness forms.

At positive lift coefficients up to about 0.5, the figure demonstrates that the NACA 64AX10 airfoil sections have essentially the same drag-divergence Mach number for varying amounts of camber corresponding to design lift coefficients ranging from 0 to 0.3. For the NACA 0010 airfoil section,large increases in the drag-divergence Mach number at moderate and large lift coefficients are found with amounts of camber up to that corresponding to a design lift coefficient of about 0.3. An amount of camber greater than that corresponding to a design lift coefficient of 0.3 leads to further gains at large lift coefficients but to losses at small and moderate lift coefficients for both basic thickness forms.

Comparison of data contained in figures 3, 4, and 5 pertaining just to the NACA 64A010 profile indicates that none of the camber lines considered produced significant increases in the drag-divergence Mach number of the NACA 64A010 airfoil section at small or moderate lift coefficients. In contrast, a similar comparison for the NACA 0010 airfoil section indicates that marked improvement in the drag-divergence Mach number characteristics of this airfoil section seems to be provided by some camber lines in common usage, such for example as the $a = 1.0$ mean line. It is thus concluded that profiles exist for which the proper choice of camber may significantly improve the high-speed drag characteristics, but that universal improvement is not to be expected.

## EXPERIMENTAL INVESTIGATION

To determine to what extent the predicted gains from the use of camber could be attained for an airfoil section considered in the preceding analysis, tests of the NACA 0010 (reference 5) and NACA 0010, $a = 1.0$, $c_{l_i} = 0.3$ airfoil sections were made in the Ames 1- by 3-1/2-foot high-speed wind tunnel. The measured section drag, lift, and pitching-moment coefficients are presented in figure 6. The models were of 6-inch chord and the test Reynolds number varied from 1 million to 2 million with increasing Mach number.

Cross plots, for various constant section lift coefficients, of the variations with Mach number of section drag coefficient, section angle of attack, section lift-curve slope, and section pitching-moment coefficient are presented in figures 7, 8, 9, and 10, respectively. Comparison with similar data for the NACA 64A010 airfoil section (reference 6) is made because this airfoil section is generally considered to have good section characteristics at high subsonic Mach numbers and thus it provides a measure of the suitability of the cambered conventional airfoil for high-speed applications.

Examination of figure 7 shows that the addition of camber was very effective in improving the drag characteristics of the uncambered basic thickness form at section lift coefficients of 0.2, 0.4, and 0.6. The magnitude of the improvement, moreover, was enough to make the section drag coefficients for the cambered conventional airfoil section as small as, or smaller than, those of the symmetrical NACA 64A010 section. The variations with Mach number for constant lift coefficient of the section angle of attack, the section lift–curve slope, and the section pitching–moment coefficient for the cambered NACA 0010 airfoil section (figs. 8, 9, and 10) are all essentially similar to those of the NACA 64A010 airfoil section throughout the range of Mach numbers for which data were obtained.

## CONCLUDING REMARKS

The results of this investigation demonstrate that a proper addition of camber to the NACA 0010 airfoil section leads to significant improvements in the drag characteristics at moderately supercritical speeds. It has been shown further that, within the test range of Mach numbers, all the high–speed aerodynamic characteristics of this cambered airfoil section are equivalent to those of the NACA 64A010 airfoil section, which is generally considered to have good high–speed characteristics.

At Mach numbers above the maximum reached in these tests, it is probable that the lift and pitching–moment characteristics of cambered NACA 0010 airfoil sections will undergo large variations such as are characteristic of other cambered sections. With some cambered sections it has been possible to reduce these variations by use of upwardly deflected plain flaps.

The preliminary analysis which has been made indicates that moderate amounts of camber and a rearward location of the position of maximum camber are most conducive to increasing the drag–divergence Mach number at moderate lift coefficients. An important limitation to this working hypothesis is that extreme rearward location of maximum camber imposes large adverse pressure gradients over the rear portion of the airfoil upper surface. Such gradients might be expected to cause boundary–layer separation and hence poor characteristics at high speeds.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif.

## APPENDIX

### Theoretical Basis for the Calculation of Drag–Divergence Mach Number

An analysis of the experimental pressure distributions over a number of airfoil sections was presented in reference 1. It was found that, for an airfoil section at a fixed angle of attack, as the free–stream Mach number is increased past the critical Mach number, the local region of supersonic flow over the airfoil increases in chordwise extent. However, the abrupt supercritical drag rise does not begin until the supersonic region envelops the airfoil crest. The airfoil crest is the chordwise location at which, for a given angle of attack, the tangent to the airfoil surface lies in the free-stream direction. After the shock wave which terminates the supersonic region moves aft of the airfoil crest the pressure distribution over the forward portion of the airfoil varies in such a manner that, with further increase in free-stream Mach number, the local Mach number at each chordwise station remains essentially constant. Thus, pressure coefficients ahead of the crest become less negative while those on the afterportion of the airfoil continue to become more negative. The resulting pressure drag is the primary cause of the abrupt supercritical drag rise. The free-stream Mach number at which this abrupt drag rise begins, the drag-divergence Mach number, is calculated by determining from the low-speed pressure distribution the free-stream Mach number at which the local velocity at the airfoil crest is sonic. The significance of local sonic velocity arises from the experimentally observed fact that, as long as there is no extensive flow separation, the terminal shock wave is located near the point on the airfoil surface at which the local pressure coefficient corresponds to sonic velocity.

## Numerical Example of Procedure for Calculating the Drag-Divergence Mach Number of an Airfoil Section

In calculating the variation of drag-divergence Mach number with lift coefficient for an airfoil section, it is convenient to determine the drag-divergence Mach number and lift coefficient for which the airfoil crest is located at various standard chordwise stations. This is a consequence of the fact that the theoretical velocity distributions for airfoil thickness and camber shapes are usually tabulated at these standard stations. To illustrate the procedure for calculating drag-divergence Mach number, consider the conditions under which the crest is at the 0.20-chord station for an NACA 0010 airfoil section with an NACA $a = 1.0$ camber line having a design lift coefficient of 0.3. The thickness distribution for the NACA 0010 airfoil section is given in reference 7. From this, the slope of the symmetrical airfoil at the 0.20-chord station can be found graphically to be 0.048. Although cambered airfoils are derived by adding the thickness distribution perpendicular to the mean line, it is sufficiently accurate to determine the slope of the upper surface of the airfoil by taking the sum of the slope of the basic thickness form and the slope of the mean line. Reference 7 contains the values of the slope of the NACA $a = 1.0$ camber line at various standard stations. For a design lift coefficient $c_{l_i}$ of 0.3 the slope at the 0.20-chord station is 0.033. Thus, for the upper surface of the cambered airfoil, the tangent to the 0.20-chord station lies in the free-stream direction when

$$\tan \alpha_o = 0.048 + 0.033 = 0.081$$

which corresponds to an angle of attack of 4.6°. Noting that for thin airfoil sections the lift-curve slope is about 0.11 and that the NACA a = 1.0 camber line attains its design lift at zero angle of attack, it follows that the low-speed additional lift coefficient (lift coefficient due to angle of attack) for the case being considered is 0.51. From reference 7, it is found that the local velocity at the 0.20-chord station for the NACA 0010 section at an additional lift coefficient of 0.51 is 1.312 times the free-stream velocity. The velocity-ratio increment due to the camber loading is 0.075,so the total local velocity is 1.387 times free-stream velocity. The corresponding pressure coefficient is

$$P = 1 - (1.387)^2 = -0.92$$

In order to determine the free-stream Mach number M at which this pressure coefficient corresponds to the occurrence of local sonic velocity, it is assumed that the pressure coefficient varies with Mach number in accordance with the Prandtl-Glauert compressibility factor. The problem then reduces to the solution of the equation

$$\frac{P}{\sqrt{1-M^2}} = \frac{2}{1.4M^2}\left[\left(\frac{2}{2.4} + \frac{M^2}{6}\right)^{3.5} - 1\right]$$

For P equal to -0.92, M is found to be 0.62. The lift coefficient at which 0.62 is the drag-divergence Mach number is then calculated to be 1.02 by correcting the low-speed lift coefficient, 0.81, by the compressibility factor.

Calculations must be made for a sufficient number of chordwise stations to define the curve of drag-divergence Mach number versus lift coefficient. Values should be computed only for points behind about the 7-percent-chord station because calculations made for stations nearer the leading edge of the airfoil are usually not in satisfactory agreement with experiment.

## REFERENCES

1. Nitzberg, Gerald E., and Crandall, Stewart: A Study of Flow Changes Associated With Airfoil Section Drag Rise at Supercritical Speeds. NACA TN 1813, 1949.

2. Göthert, B.: High Speed Tests in the DVL High-Speed Wind Tunnel (2.7 m ∅) on Aerofoils of the Same Thickness Distribution With Different Cambers (Series NACA f 35 12-0, 55 40). British Ministry of Supply GDC 10/1408 T. (Reports and Translations No. 400) May 1947.

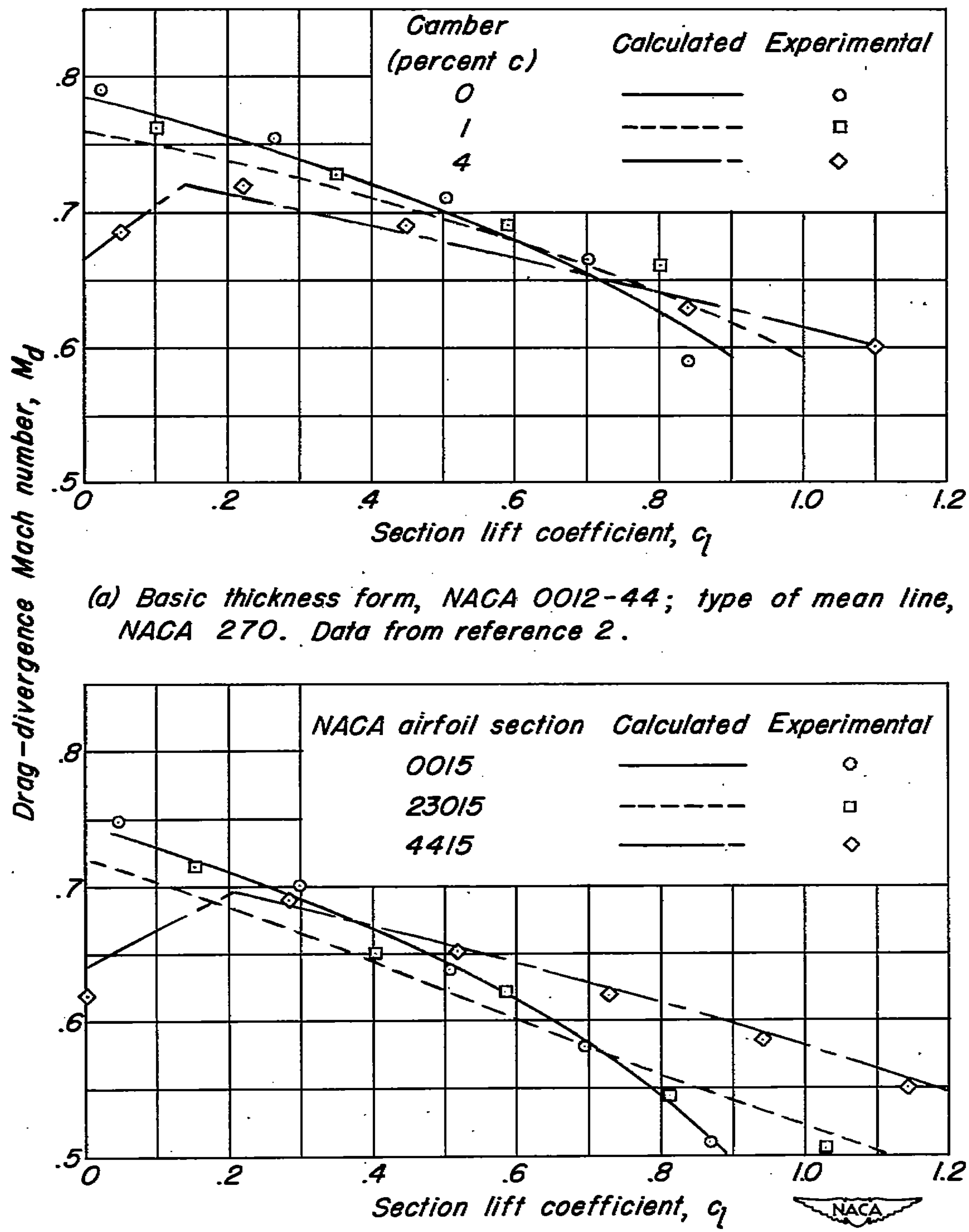
3. Graham, Donald J., Nitzberg, Gerald E., and Olson, Robert N.: A Systematic Investigation of Pressure Distributions at High Speeds Over Five Representative NACA Low-Drag and Conventional Airfoil Sections. NACA Rep. 832, 1945.

4. Lindsey, W. F., Stevenson, D. B., and Daley, Bernard N.: Aerodynamic Characteristics of 24 NACA 16-Series Airfoils at Mach Numbers Between 0.3 and 0.8. NACA TN 1546, 1948.

5. Polentz, Perry P.: Comparison of the Aerodynamic Characteristics of the NACA 0010 and 0010-64 Airfoil Sections at High Subsonic Mach Numbers. NACA RM A9G19, 1949.

6. Hemenover, Albert D.: Tests of the NACA 64-010 and 64A010 Airfoil Sections at High Subsonic Mach Numbers. NACA RM A9E31, 1949.

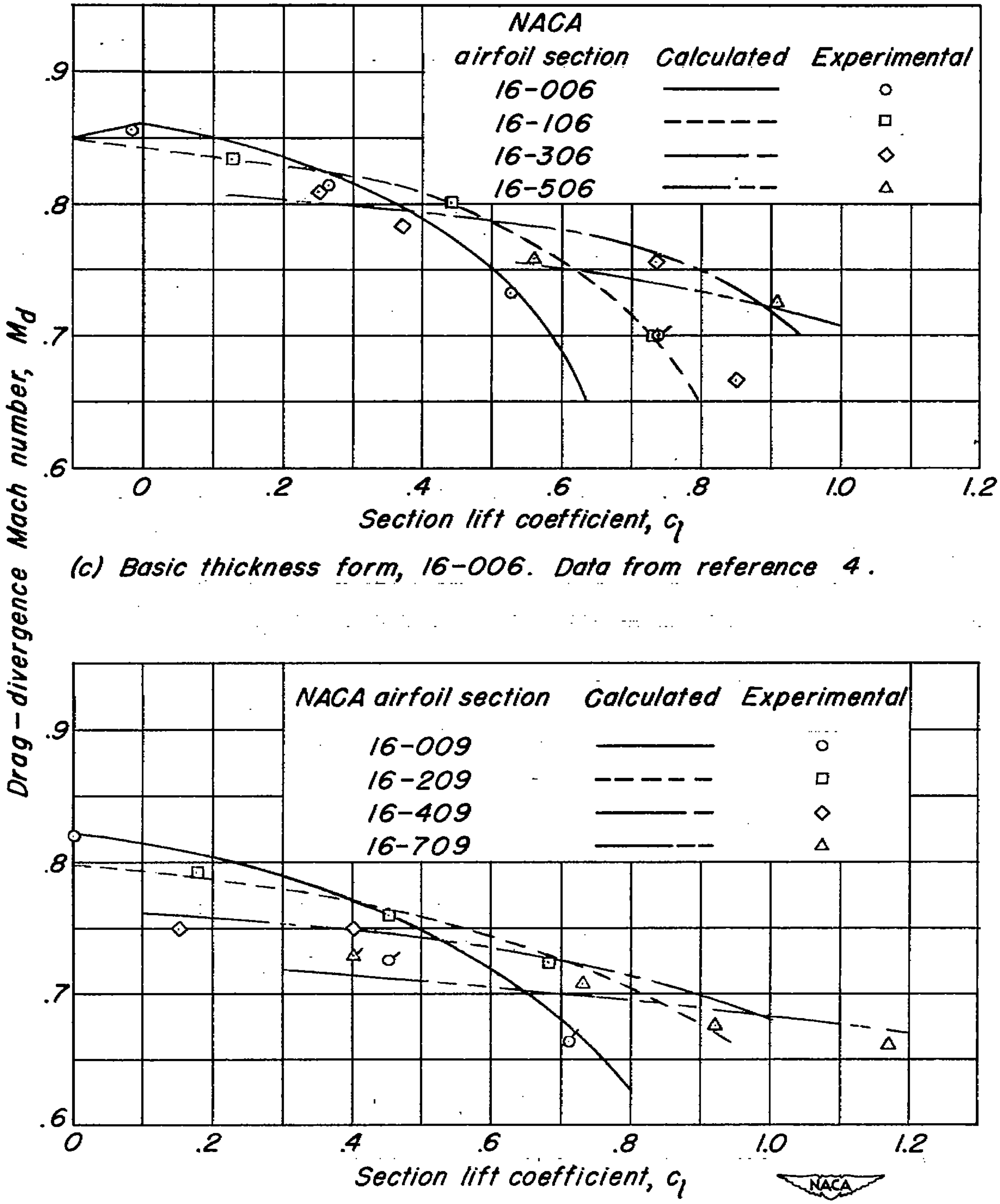
7. Abbot, Ira H., von Doenhoff, Albert E., and Stivers, Louis S., Jr.: Summary of Airfoil Data. NACA Rep. 824, 1945.




(a) Basic thickness form, NACA 0012-44; type of mean line, NACA 270. Data from reference 2.

(b) Basic thickness form, NACA 0015. Data from reference 3.

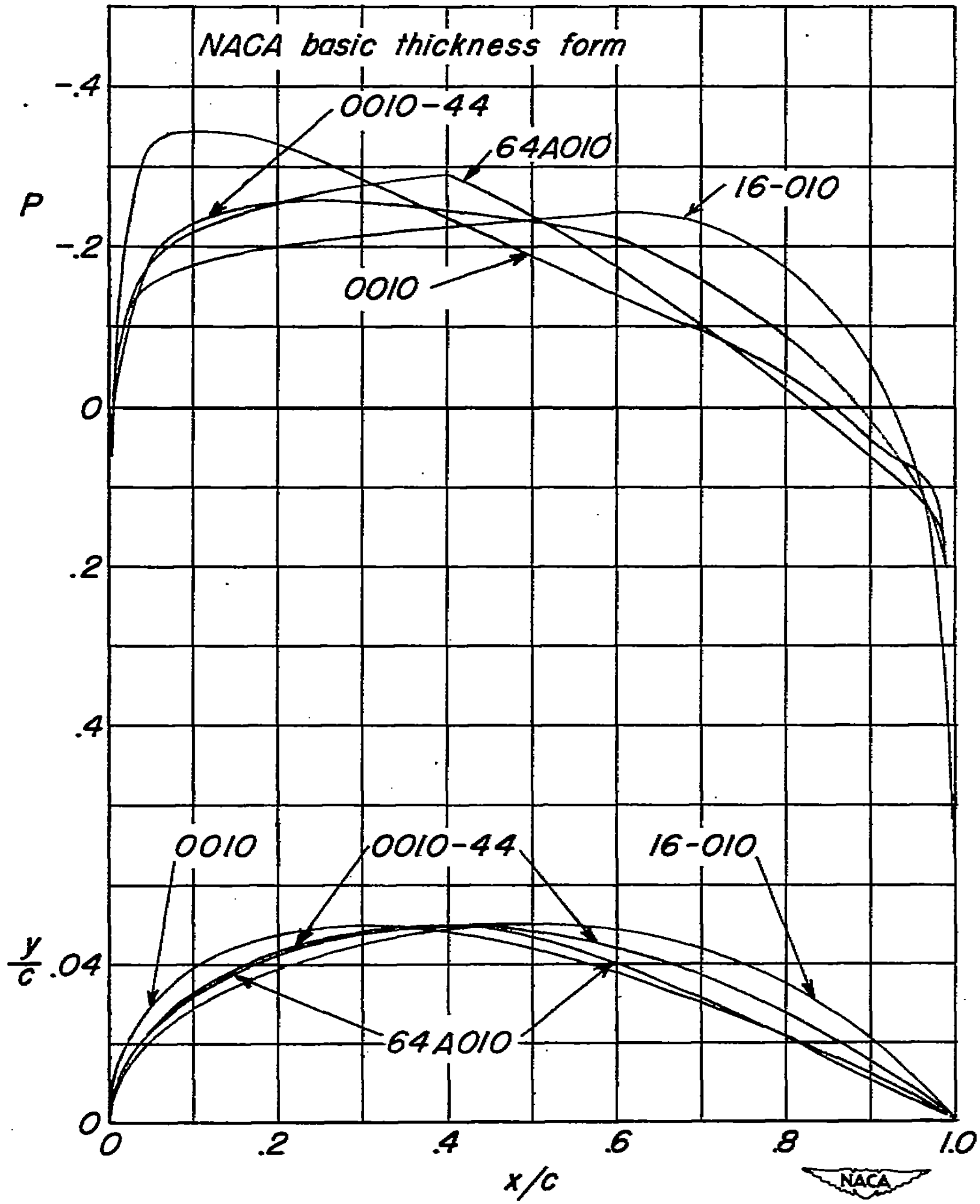
Figure 1.– Effect of camber on the variation of drag-divergence Mach number with section lift coefficient for several basic thickness forms.




(c) Basic thickness form, 16-006. Data from reference 4.

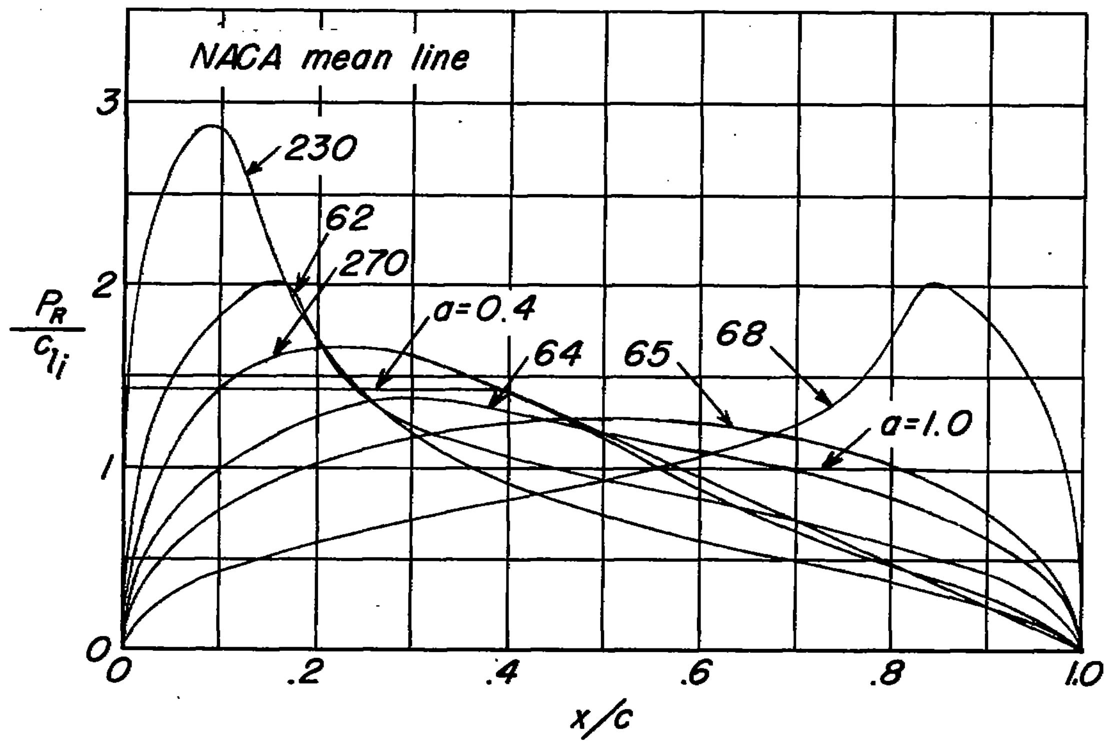
(d) Basic thickness form, 16-009. Data from reference 4.

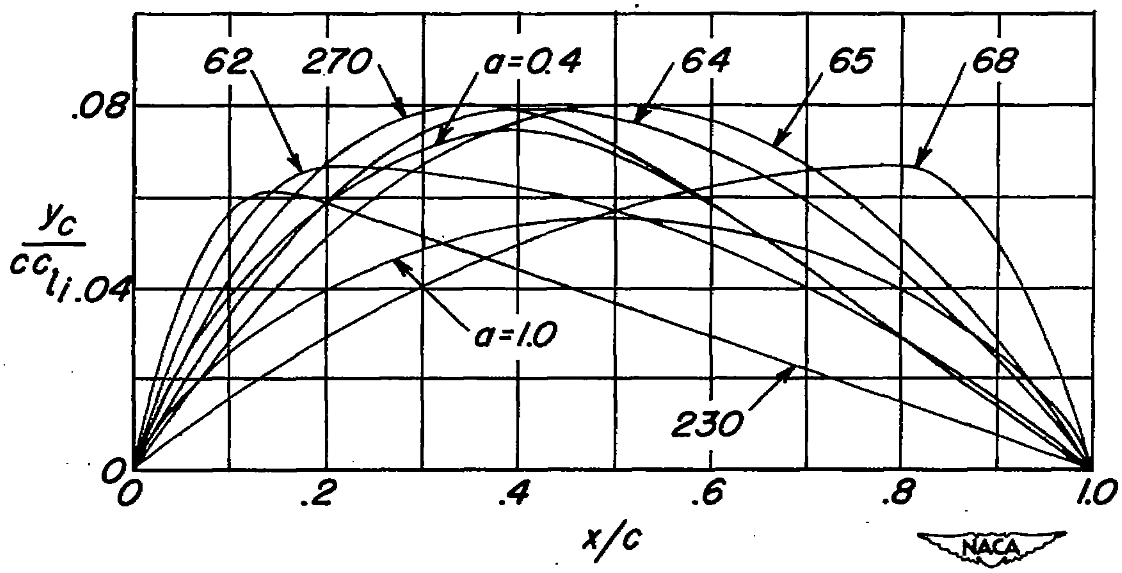
Figure 1.–Concluded.




(a) Basic-thickness-form ordinates and pressure distributions.

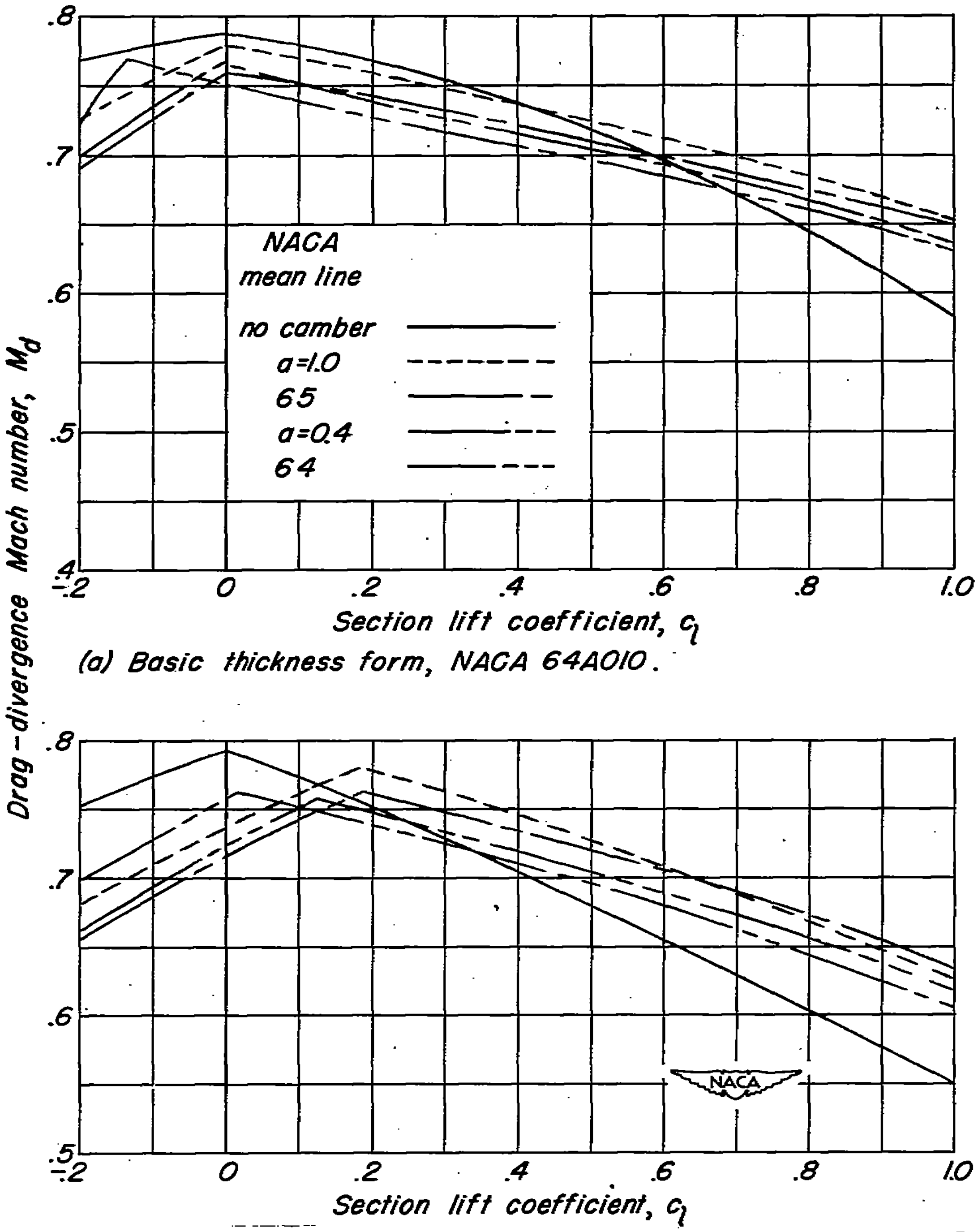
Figure 2.– Shapes and pressure distributions of the various airfoil–section mean lines and thickness distributions considered.




(b) Mean lines and mean-line load distributions.

Figure 2.–Concluded.




(a) Basic thickness form, NACA 64A010.



(b) Basic thickness form, NACA 0010.

*Figure 3.– Effect of type of mean line on calculated variation of drag-divergence Mach number with section lift coefficient. $c_{l_i}$=0.3.*

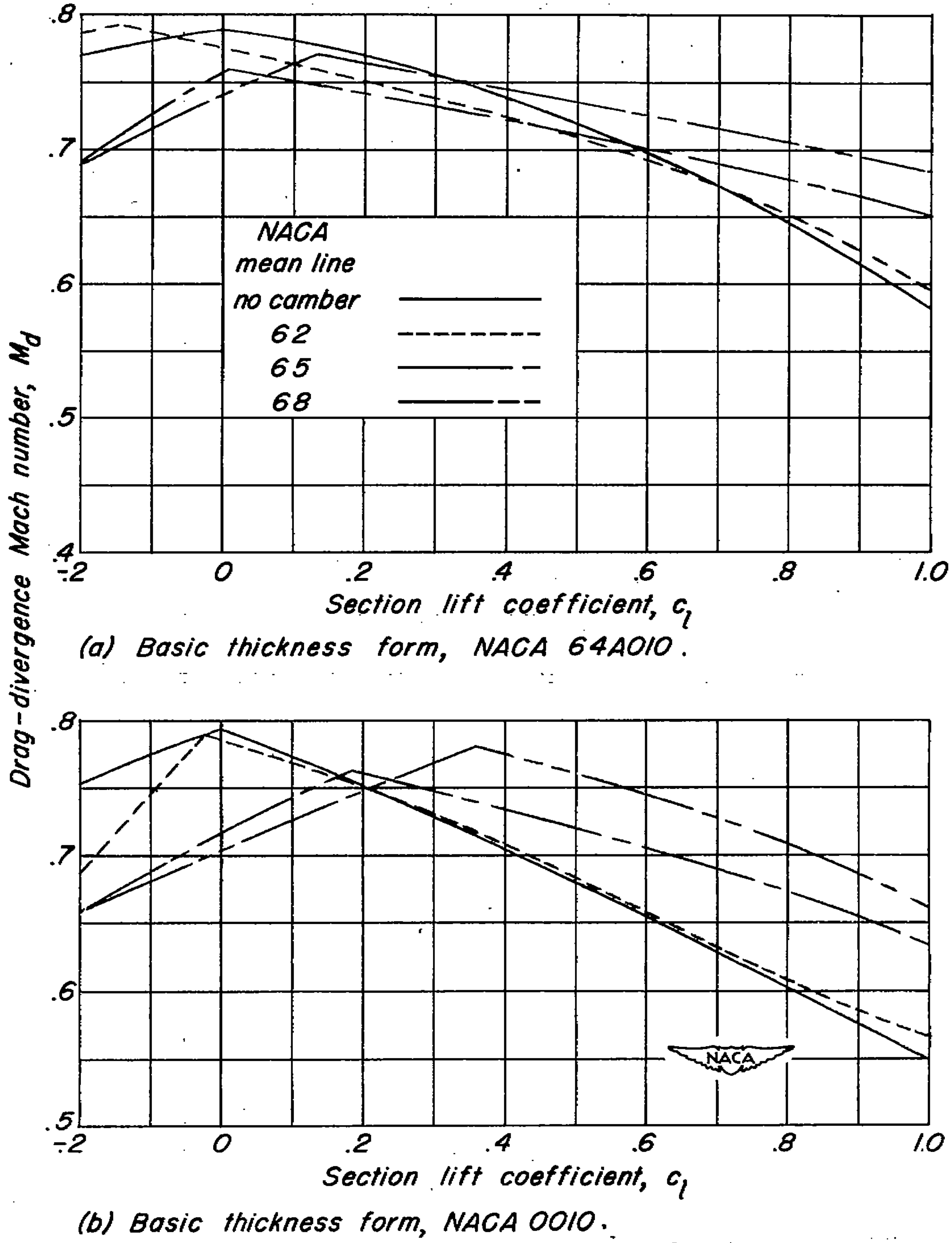
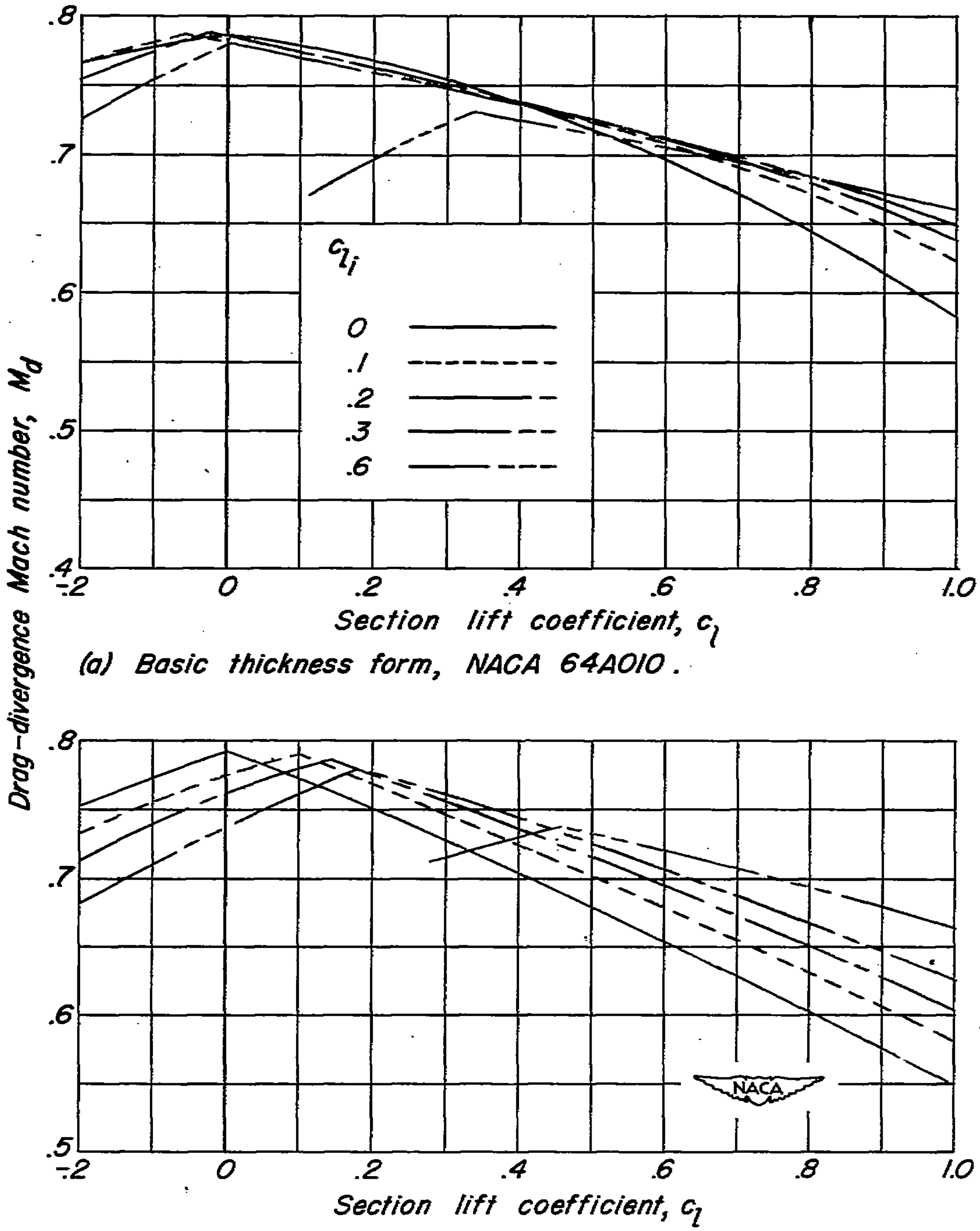


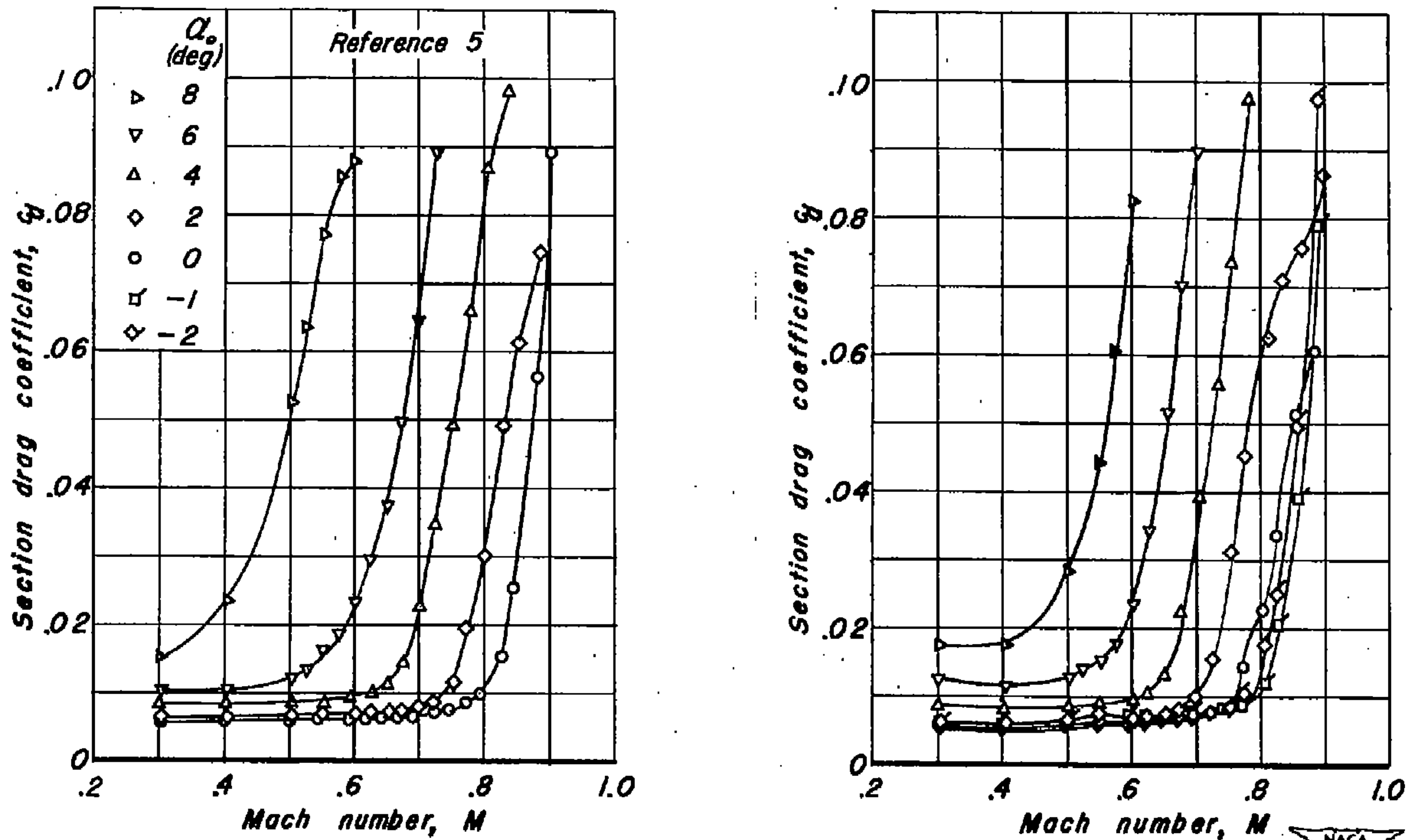

*Figure 4.- Effect of chordwise location of maximum camber on calculated variation of drag-divergence Mach number with section lift coefficient. $c_{l_i}$=0.3.*




(a) Basic thickness form, NACA 64A010.

(b) Basic thickness form, NACA 0010.

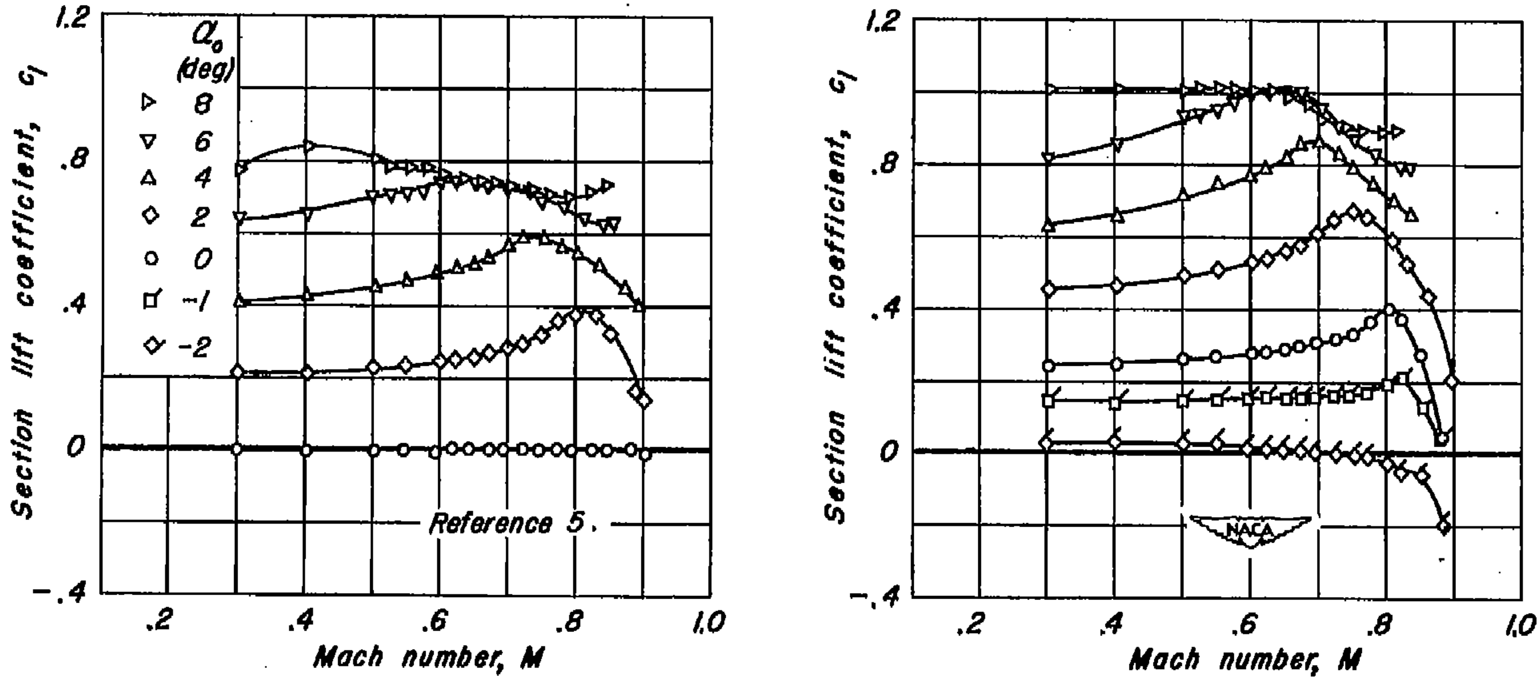
*Figure 5.- Effect of design lift coefficient on calculated variation of drag-divergence Mach number with section lift coefficient for airfoil sections with a=1.0 type mean lines.*




NACA 0010 NACA 0010, $a=1.0$, $c_{l_i}=0.3$

(a) Variation of section drag coefficient with Mach number.

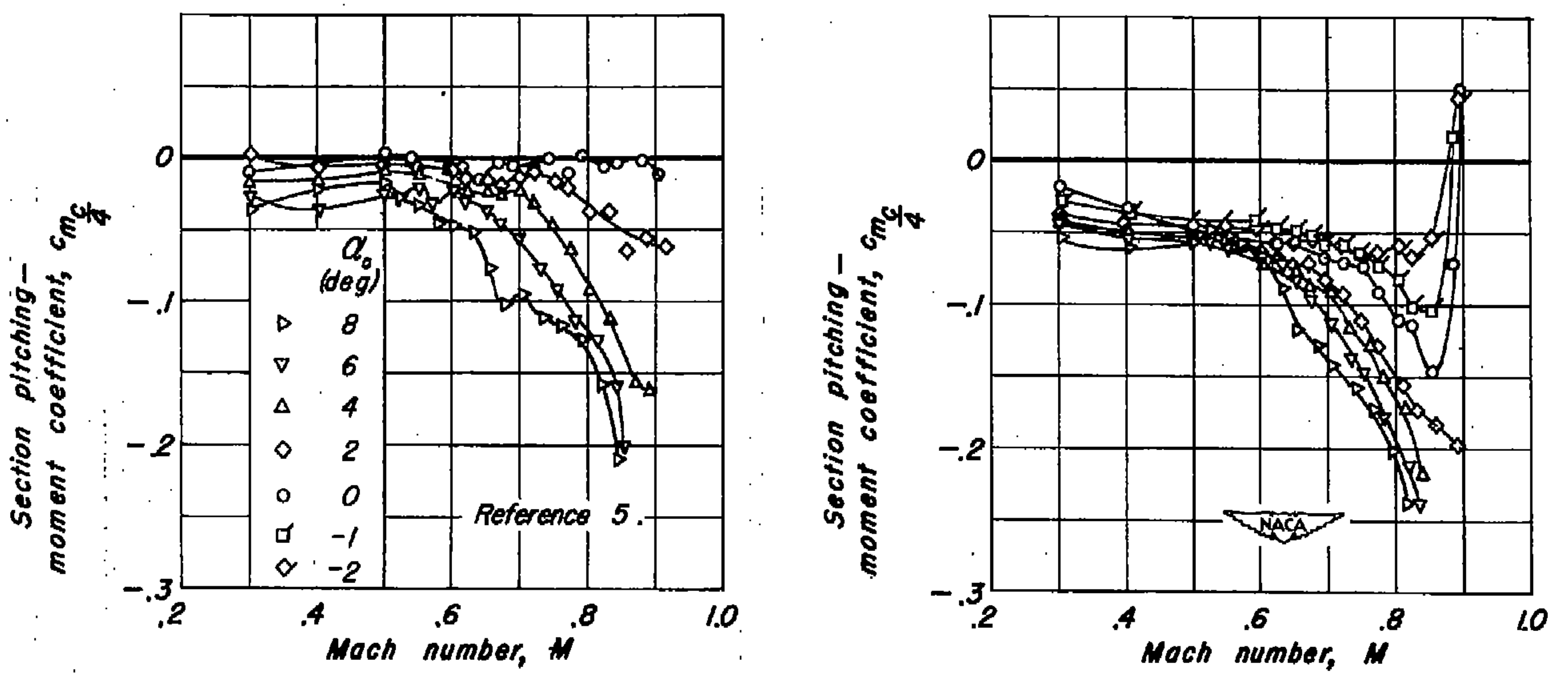
Figure 6.– Measured variation of section force coefficients with Mach number for NACA 0010 and NACA 0010, $a=1.0$, $c_{l_i}=0.3$ airfoils.




NACA 0010          NACA 0010, $a=1.0$, $c_{l_i}=0.3$

(b) Variation of section lift coefficient with Mach number.

Figure 6.—Continued.




NACA 0010

NACA 0010, a=1.0, $c_{l_i}$ = 0.3

(c) Variation of section pitching-moment coefficient with Mach number.

Figure 6.— Concluded.

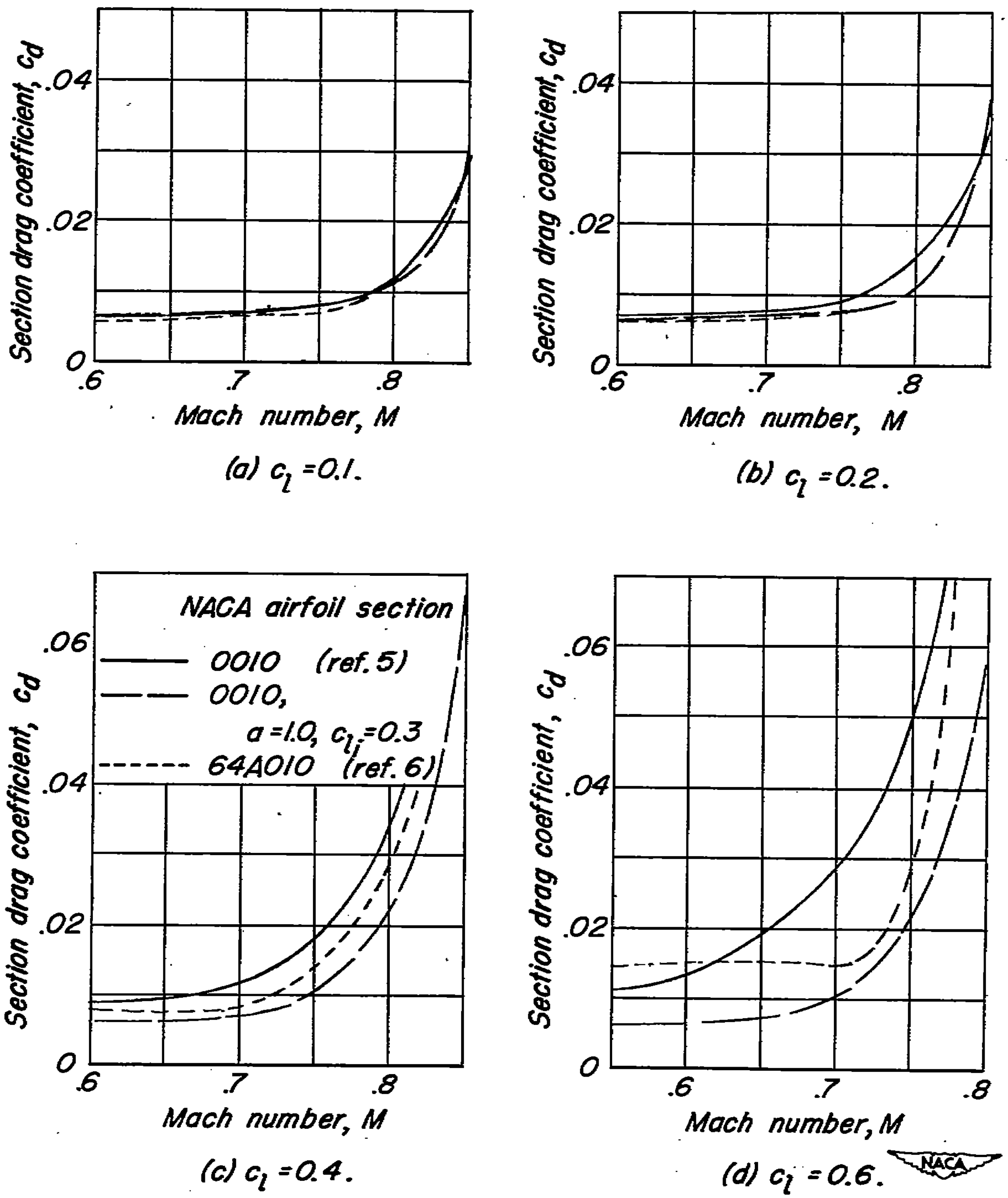



*Figure 7. – Variation of section drag coefficient with Mach number at constant section lift coefficient.*

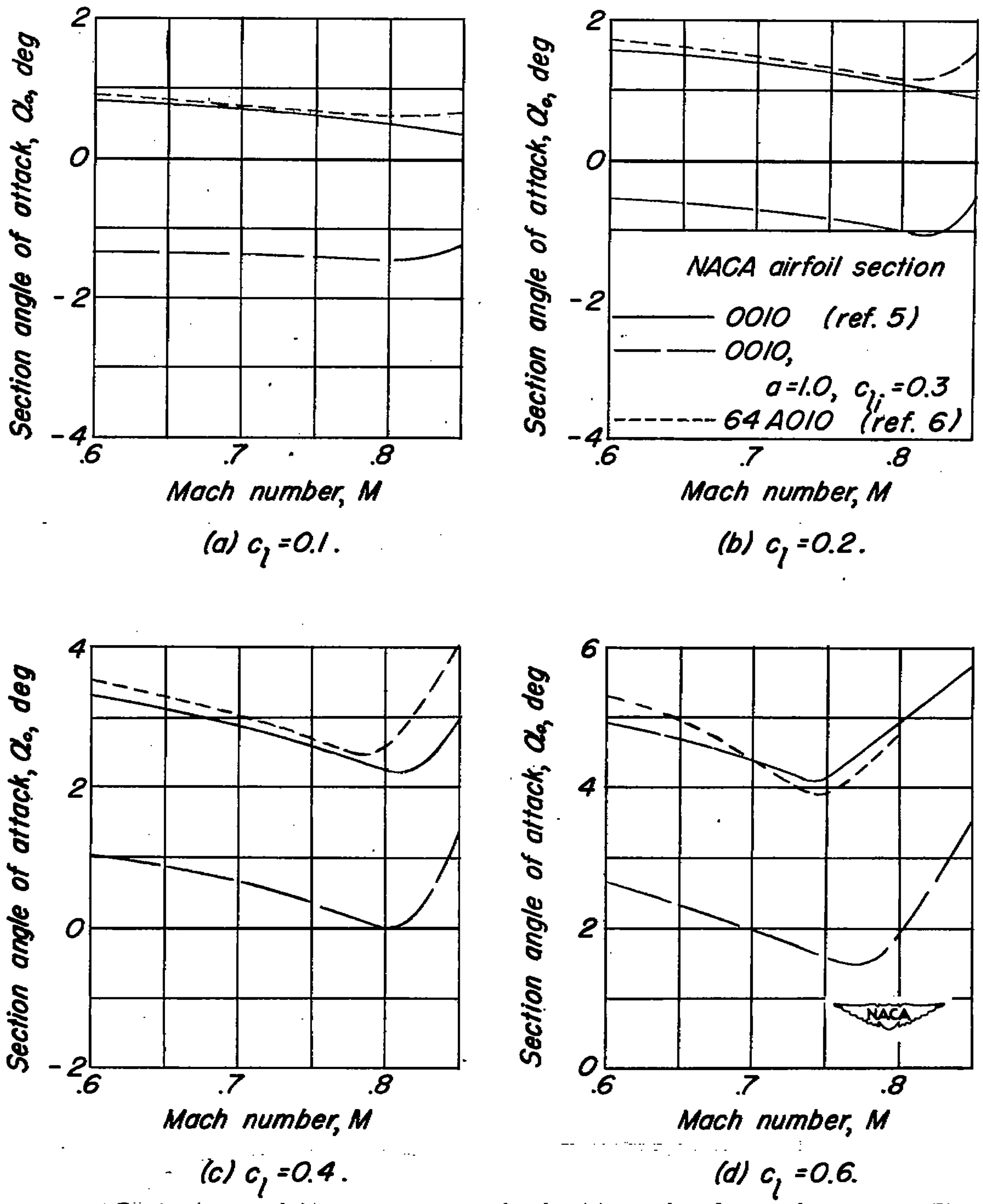



*Figure 8.– Variation with Mach number of section angle of attack at constant section lift coefficient.*

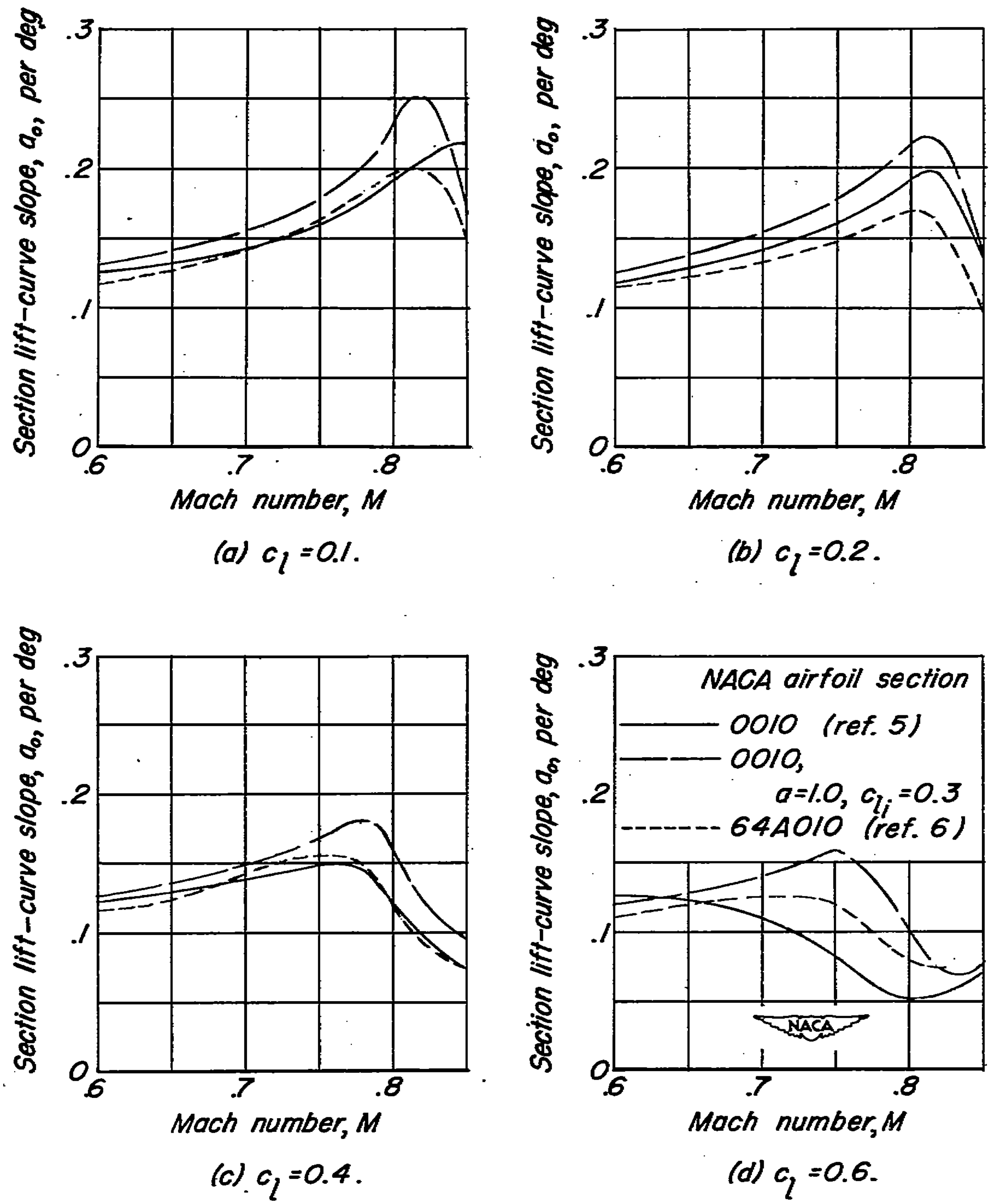
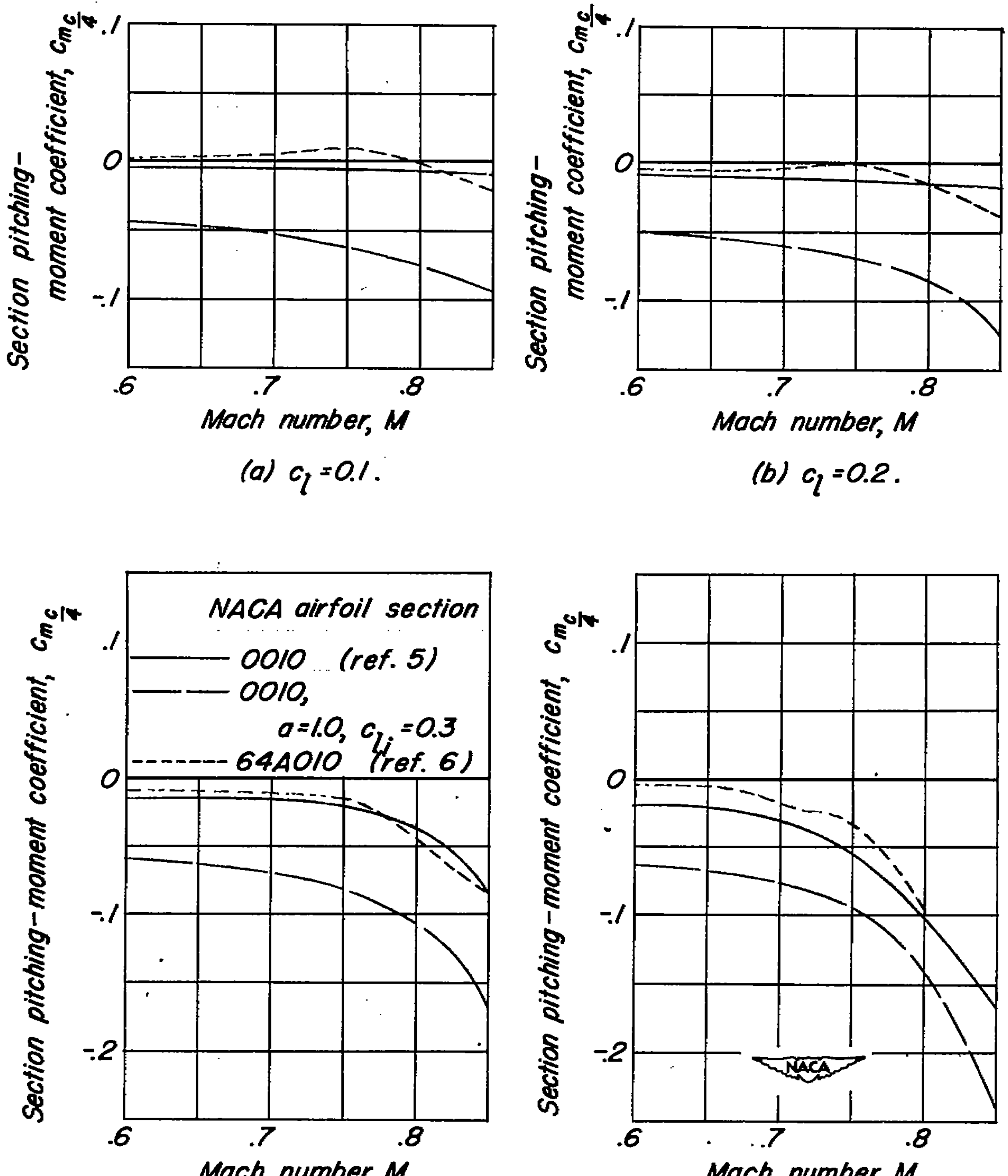



*Figure 9.– Variation of section lift-curve slope with Mach number at constant section lift coefficient.*




(a) $c_l = 0.1$. (b) $c_l = 0.2$.

(c) $c_l = 0.4$. (d) $c_l = 0.6$.

*Figure 10.– Variation of section pitching-moment coefficient with Mach number at constant section lift coefficient.*